

**GUGGENHEIM AERONAUTICAL LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY**



HYPERSONIC RESEARCH PROJECT

Memorandum No. 41

November 1, 1957

**MEASUREMENT OF TURBULENT HEAT TRANSFER RATES
ON THE AFT PORTION AND BLUNT BASE
OF A HEMISPHERE-CYLINDER IN THE SHOCK TUBE**

by

Josef Rabinowicz

ARMY ORDNANCE CONTRACT NO. DA-04-495-Ord-19

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SUMMARY

Turbulent heat transfer rates on the aft portion and on the blunt base of a hemisphere cylinder were measured in the 2-7/8" x 2-7/8" GALCIT shock tube over a range of shock Mach numbers between 3.25 and 5.1 and initial pressures between 3 and 17 cm. Hg. The local Reynolds numbers on the cylindrical afterbody varied between 3.5×10^4 and 3.0×10^5 per cm. A side support was used for the model in order to eliminate the disturbing effect of a rear sting support.

The measured turbulent heat transfer rates on the cylindrical portion agreed very well with previous flat plate measurements for small temperature differences, although the ratio of stagnation to surface enthalpy varied between 3 to 8 in the present tests. Only a slight effect of this large variation in h_s / h_w was detected in this range of local Mach numbers, i. e., $1.25 < M_e < 1.5$. The measured heat transfer rate on the base indicated that at the center of the base the heat transfer rate is comparable to that on the surface just ahead of the base, while the heat transfer rate falls off to 1/2 to 1/3 of this value towards the rim of the base. This unexpected distribution of heat transfer rate over the base, and particularly the high value at the center, shows the necessity for a careful study of wake phenomena.

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LIST OF SYMBOLS

C_H	Stanton number = $\frac{\dot{q}}{\rho_e u_e (h_{aw} - h_w)}$
h	enthalpy, cal./gr.
M	Mach number
$Nu_{\bar{x}}$	Nusselt number = $C_H \cdot Pr \cdot Re_{\bar{x}}$
$Re_{\bar{x}}$	Reynolds number = $\frac{\rho_e u_e \bar{x}}{\mu_e}$
Pr	Prandtl number = $\frac{c_p \mu_e}{k_e}$
R	body radius, cm.
p	pressure, cm. Hg. abs.
\dot{q}	heat transfer rate, cal./cm. ² /sec.
r	recovery factor
u	velocity, cm./sec.
δ	boundary layer thickness, cm.
ρ	density, gr./cm. ³
μ	viscosity coefficient, gr./cm.-sec.

Subscripts

e	value at the outer edge of the boundary layer
1	ahead of shock front
2	behind shock front
s	stagnation
aw	recovery
w	wall

I. INTRODUCTION

The use of the shock tube to simulate various aspects of hypersonic flight is now widely accepted. Recent development of the film resistance thermometer has made possible accurate heat transfer studies in the extremely short testing times available in the shock tube. Measurements of laminar heat transfer rates at and near the stagnation region of a blunt-nosed body (References 1 and 2) indicated a very good agreement with the equilibrium laminar boundary layer theory for hypersonic flight speeds as developed by Lees (Reference 3) and Rose and Riddell (Reference 4). The present work extends these measurements to the aft-region of a blunt body, i. e., to the aft surface and to the model base. For reasons which will be explained in detail in Section III, it is very difficult to obtain undisturbed laminar boundary layer flow on the aft portion of a long model in the present straight section of the 2-7/8" x 2-7/8" shock tube. Thus the work was confined to turbulent heat transfer studies.

Published experimental data on turbulent heat transfer rates with large temperature differences across the boundary layer are scarce. In a recent report, Libby and Cresci (Reference 5) compare their turbulent heat transfer rate data measured on a blunt nosed body in a wind tunnel with several analyses and find the data to agree best with the local flat plate approximation. However, they suggest that their result should be substantiated at a higher stagnation enthalpy, since their wind tunnel tests were limited to ratios of stagnation enthalpy to surface enthalpy of 1.3 to 2.0. The shock tube enables us to realize much higher ratios of stagnation to surface enthalpy, and in the present

work this ratio lies between 3 and 8. The results of the present investigation tend to confirm the previous conclusion of the validity of the flat plate analysis. This result, which is at first somewhat surprising, fits well into the overall trend indicating no startling or unexpected phenomena associated with high temperatures at ordinary pressures.

The heat transfer problem on a blunt base is considerably more difficult. Much experimental and theoretical work has been done on the base pressure problem from the point of view of an overall "mixing" analysis. However, the heat transfer rate probably depends also on local conditions, so that a knowledge of the flow field near the base surface is required before local heat transfer rates can be predicted. The measurement of base heat transfer rates in the shock tube is also complicated by the question of the time required to establish a steady wake behind a model. Thus the present experiment has two main objectives: (1) to determine whether time-independent heat-transfer rates on a blunt base are established in the time interval available in the shock tube (about 400 - 600 μ sec.); (2) to investigate heat transfer rates on a blunt base when the body is supported from the side, in order to eliminate the usual disturbing effect of a rear sting support.

The shock tube and the instrumentation used in this investigation are described in Reference 1; therefore, only a brief summary will be included in Section II. Section III presents the experimental results, including a discussion and indication of future work on this problem.

The author would like to thank Professor L. Lees for suggesting the problem and for his many helpful discussions and suggestions.

II. EXPERIMENTAL TECHNIQUE

A. Shock Tube and Instrumentation

This investigation was performed in the 2-7/8" x 2-7/8" GALCIT shock tube (Figure 1). Heat transfer rate and schlieren studies were carried out with the model placed at the 19' station of the straight section. The instrumentation block diagram is shown in Figure 2. (See Reference 1.) Shock speed is determined by measuring the time for the shock to travel between two monitoring stations spaced 2' apart. The heat transfer gages are used for detection, and the time is measured by the 7360 Berkeley counter. The schlieren spark is triggered through a time delay, which is activated by the downstream wave-speed monitoring gage.

The heat transfer gages are sputtered directly on the model. Their initial resistance was measured by a Wheatstone bridge and the current was measured by a milliammeter. The gage output is fed through a Tektronix 121 wide band amplifier to a 535 Tektronix oscilloscope, and the records are obtained by a polaroid camera.

The gages are calibrated for quantitative heat transfer measurements utilizing the method described in Reference 1. The overall accuracy of the heat transfer instrumentation is about ± 5 per cent as demonstrated in Reference 1.

B. The Hemisphere-Cylinder Model

The model consists of a brass cylinder 3/4" in diameter and 2-5/8" long, with a hemispherical nose. A 3/16" spike 3/4" long is attached to the nose in order to produce a separated region and thus

insure a fully-developed turbulent boundary layer upon reattachment of the flow to the cylinder surface (Figure 3). The aft portion of the model consists of a hollow pyrex glass "cup" $3/4$ " diameter and 1" long, which is waxed on the mating brass cylinder. The outside diameter of the glass and brass parts are matched to provide a continuous smooth surface, and the platinum film resistance gages are sputtered on the glass "cup" (Figure 3). Four gages are positioned on the flat base -- No. 1 at the center and Nos. 2, 3, and 4 on the circumference of a $1/2$ " circle as shown in Figure 4. Three additional gages are positioned on the cylindrical surface .082", .442", and .772", respectively, from the base, and are denoted as gages Nos. 5, 6, and 7 on Figure 4.

The model is side-supported by a 45° swept-back, double wedge that is $3/4$ " wide and 0.100" thick. This support enables schlieren observation of the aft-part of the model and portions of the wake, and eliminates the need for a rear-sting.

The leads on the glass "cup" are provided by a thick layer of silver paint to which wires are soldered on the unexposed edge of the glass "cup" (Figure 3). The wires are then led through machined grooves in the model surface, then along the side support to the outside of the shock tube. The "cup" is aligned so that gages Nos. 3, 5, 6, and 7 are opposite the wake of the side support in order to minimize the effect of this wake on the readings of these gages (Figures 3 and 4).

III. EXPERIMENTAL RESULTS

A. Range of the Experiments

A preliminary investigation was undertaken to determine the maximum size of the heat transfer model permitted by the blockage effects -- specifically by the effect of the reflection of the bow shock wave from the shock tube wall. It was determined that a $3/4$ " diameter body can be accommodated in the $2-7/8$ " x $2-7/8$ " section without choking the flow. For this model the ratio of blocked area to the tube cross section area is .07, including the projected area of the side support. However, the flow Mach number is only 1.3 to 2.0, which presents a more difficult problem so far as model length is concerned. The bow wave angle is relatively steep, and the first shock reflection from the shock tube wall reaches the model at a station between $3/4$ " and $1-1/2$ ", measured from the model nose. Thus it is impractical to attempt to test a model which is shorter than the distance of the first shock reflection from the nose. The model was designed to extend about 1" behind the first shock reflection, thus anticipating a range of shock strengths within which the second shock reflection from the tube wall will be weak enough and will strike the wake far enough downstream of the base so as to have little or no effect on the wake structure and the base heat transfer rate.

The flow around the actual heat transfer model is shown in the schlieren photographs in Figure 5. Figures 5a and 5b show the developing flow at $\Delta t = 40 \mu\text{sec.}$ and $140 \mu\text{sec.}$, respectively, for a shock Mach number M_s of 3.8. Figure 5c shows the flow at $\Delta t = 350 \mu\text{sec.}$ for $M_s = 3.0$. Figure 5b indicates a fully-developed wake and a relatively weak second reflection, while at the lower Mach number (Figure 5c)

the flow is obviously choked. At high Mach number $M_s > 4.5$ the reflected shocks again increase in strength, and a choking effect again appears. Thus the base heat rate measurements were limited to a narrow range of shock Mach numbers ($3.5 < M_s < 4.0$) because of the particular combination of flow Mach number and test section geometry. The initial pressures were varied between 3 and 17 cm. Hg. corresponding to Reynolds numbers/cm. in the range 3.5×10^4 to 3×10^5 . The lower limit is imposed by the minimum pressure required for a fully-developed turbulent boundary layer, and the upper limit is set by structural considerations. The advantage of larger tube cross section dimensions and of higher flow Mach number are obvious. These considerations indicate the advantages of using an expansion nozzle as the test area for this type of an investigation.

The limitations on the range of shock Mach numbers for the turbulent heat rate measurements on the cylindrical surface are not as severe, because the expansion around the shoulder limits the feedback from the wake. Good results are obtained for shock Mach numbers in the range of 3.25 to 5.1.

The disturbance caused by the shock wave reflection on the model surface ahead of the heat transfer gages causes disturbances in the laminar boundary layer when one attempts to obtain laminar heat rate measurements. However, the effect is negligible for turbulent boundary layers, so the present investigation was confined to the turbulent case.

B. Turbulent Heat Transfer Rate Measurements on the Cylindrical Surface

Representative gage outputs at various positions on the model are shown in Figure 6. Figure 6d is a record of gage No. 7 on the surface of the cylinder. Measured heat transfer rates from gages Nos. 5, 6, and 7 at $M_s = 5.8$ are shown in Figure 7 as a function of initial pressure p_1 . The lines drawn show a relation $q \sim p_1^{0.8}$.^{*} The heat transfer rate measured by gage No. 1 on the base is also shown for comparison. These base measurements will be discussed in Section III. C.

In Figure 8 the heat transfer rates measured at the various locations on the cylindrical surface of the model are plotted as $\dot{q}/p_1^{0.8}$ vs. M_s (where p_1 is measured in cm. Hg. abs. and \dot{q} is cal./cm.²/sec.). For comparison the stagnation point heat transfer rate for $p_1 = 1$ cm. Hg. abs. calculated for the equilibrium laminar boundary layer according to Reference 3, is also shown. Turbulent heat transfer rates on the cylindrical afterbody are 1/4 to 1/3 of the laminar stagnation point heat transfer rate for comparable free stream conditions.

Turbulent heat transfer rates are generally expressed in terms of the Stanton number, C_H , where

$$\dot{q} = \rho_e u_e C_H (h_{aw} - h_w) . \quad (1)$$

Semi-empirical correlations of the available experimental data give

$$C_H = B Re^a Pr^b \quad (2)$$

* The actual Mach number varies between 3.5 and 4.0 and the heat transfer rates are normalized to $M_s = 3.8$ by the factor $[M/(\bar{M}_s)]^3$.

where $a = -0.2$, $b = -2/3$, and B depends on the local Mach number and the temperature (or enthalpy) ratio across the boundary layer. Also

$$h_{aw} = h_{se} \left[1 - (1 - r) \frac{u_e^2}{2h_{se}} \right], \quad \text{where the recovery factor } r \text{ is}$$

approximately $Pr^{1/3}$, i. e., $r = 0.89$ for $Pr = 0.7$. According to Eqs. (1) and (2),

$$\dot{q} = B (\rho u)_e^{0.8} \mu_e^{0.2} (Pr)^{-2/3} (h_{aw} - h_w) (\bar{x})^{-0.2}, \quad (3)$$

where \bar{x} is some effective length of run for the fully-developed turbulent boundary layer. In other words in the present experiments the quantity

$$\frac{\dot{q} \bar{x}^{-0.2}}{P_1^{0.8}} \quad \text{should be the same for all three gage locations at the same}$$

shock Mach number. This effective distance \bar{x} is determined directly from the measured values of \dot{q} , and turns out to be remarkably constant for all three gages over the range $3 < M_s < 5.1$, with the following

values: $\bar{x}_7 = 0.50''$, $\bar{x}_6 = 0.82''$, $\bar{x}_5 = 1.19''$ within ± 5 per cent.* In

Figure 9 the quantity $\frac{\dot{q} \bar{x}^{-0.2}}{P_1^{0.8}}$ is plotted against shock Mach number;

evidently the "spread" between the three gages shown in Figure 8 is practically eliminated.

Previous measurements of turbulent heat transfer rates with zero pressure gradient are correlated very well by Eq. (3), where

* This effective distance \bar{x} for any gage is very nearly equal to the distance measured from the point on the model at which the bow shock reflected from the shock tube wall strikes the model surface (Figure 5).

$B = 0.03$ for $M_e \cong 0$ with $T_w/T_e \cong 1$; $B = .0266$ for $M_e = .87$ and $B = .021$ for $M_e = 1.62$, with $T_w/T_{aw} \cong 1$, i. e., small temperature differences. These data are obtained from References 6, 7, 8, 9, and 10 and are reproduced in Figure 10. In order to compare the present results with this earlier work the "external" flow quantities $(\rho u)_e$ and μ_e for the present experiments were computed by assuming a normal shock ahead of the model nose and an isentropic expansion along the body surface from the stagnation point to the free stream pressure, p_2 . The turbulent heat transfer data obtained in the shock tube agree remarkably well with the earlier data of References 6, 7, and 8 in which $T_w/T_{aw} \sim 1$. In Figure 9 the curves representing the average of these data (obtained from Figure 10) at Mach numbers $M = 0$, $M = 0.87$, and $M = 1.62$ are shown. The shock tube data which corresponds to local Mach numbers between 1.25 and 1.5 fall between the curves for $M = 0.87$ and $M = 1.62$. The remarkable fact here is that the corresponding ratio of stagnation to surface enthalpy increases from 3 to 8 in the present tests.

When one plots the data in the form $Nu_{\bar{x}}$ vs $Re_{\bar{x}}$ disregarding the variation in enthalpy level (Figure 11), the scatter in the data increases slightly as compared to Figure 9. This scatter indicates a slight effect caused by the variation of h_s/h_w . One can conclude that in this Mach number range* the effect of enthalpy level is surprisingly small. This conclusion is in agreement with the previous results of Libby and Cresci (Reference 5).

* This result does not imply that the effect of h_s/h_w is small also at high local Mach numbers.

The experiments of Libby and Cresci (Reference 5) can also be used to observe the effect of falling pressure gradients on turbulent heat transfer rates. These experiments were performed in a blow-down wind tunnel with varying stagnation pressures and temperatures during the run. In order to correlate the accumulated data, modified Reynolds and Nusselt numbers, based on stagnation conditions, were used. It is difficult without recomputing the data to obtain local heat transfer values. However, a comparison with flat plate analysis is presented in Reference 5, from which the agreement with the local flat plate data can be examined.* The geometrical distance from the stagnation point has been used in evaluating the Reynolds and Nusselt numbers. This procedure is consistent with most turbulent boundary layer studies on flat plates. The fact that it seems to give good results in the case of an axially symmetric body with a falling pressure gradient is worth noting. Figure 7 of Reference 5 shows that the measured heat transfer rates fall within ± 10 per cent of the curve described by Eq. (6) of Reference 5. Translating the particular definition of the modified Reynolds and Nusselt numbers into their conventional form results in the following relation

$$\text{Nu}_x = 0.03 (\rho'/\rho_e)^{0.8} (\mu'/\mu_e)^{0.2} (\text{Pr})^{1/3} (\text{Re}_x)^{0.8} \quad (3)$$

where ρ' and μ' are defined at reference conditions determined by the local pressure and reference enthalpy

* It may be that some of the scatter of the experimental results may be due to the correlation with respect to stagnation rather than local conditions. A replotting of the data in terms of the conventional local quantities would be very helpful in generalizing these data to other heat transfer applications.

$$h' = 0.5 h_w + 0.22 h_{s_e} + 0.28 h_e \quad (4)$$

Now for Libby and Cresci experiments h_s/h_w varies between 1.3 and 1.7, thus $h' \cong h_e$. The local Mach number at the thermocouple positions in these experiments varied between 0.5 to 0.8. Thus one observes that these data agree with the flat plate data with no pressure gradients at the corresponding Mach numbers.

One can conclude from the collection of data presented here that for subsonic and low supersonic flow Mach numbers, the turbulent heat transfer rates seem to be to a surprising degree independent of falling pressure gradients, stagnation to wall enthalpy ratio, and radius of curvature when $\delta/R \ll 1$. The only appreciable effect is found to be the local Mach number, which is responsible for about 15 per cent reduction in the heat transfer coefficient by the increase of local Mach number from 0 to about 1.5.

C. Turbulent Base Heat Transfer Measurements

The arrangement of the four heat transfer gages on the blunt base is shown in Figure 4, and representative outputs of gages Nos. 1, 3, and 4 are shown in Figures 6a, 6b, and 6c. The output of these gages is not as smooth as that of the surface gages Nos. 5, 6, and 7. However, the output of gages Nos. 1 and 3 indicates a steady heat transfer rate after a "building-up" period of about 100 - 150 μ sec. Gage No. 4 lies in the wake of the side support and the output of this gage never gave a steady heat transfer rate. The comparison of the base heat rate at the center and at a location 1/4" from center as measures with gages Nos. 1 and 3, is shown in Figure 8. Gage No. 1 at the center of the base indicates a heat rate comparable with

that of gage No. 5 on the surface just ahead of the base. At gage No. 3 position the heat rate is only $1/2$ to $1/3$ of this value. One should note that the output of gage No. 4 (though disturbed) is comparable to that of gage No. 3, which increases the confidence in this result.

The experimental data indicate that a "steady" heat transfer rate is established after about $100 - 200 \mu\text{sec}$. However, the total testing time is about $400 - 600 \mu\text{sec}$., which may not be long enough to establish steady equilibrium wake flow such as may be experienced in a wind tunnel or in free flight. Until this question is settled the present base heat transfer data is considered to be preliminary and is reported mainly to call attention to this result and stimulate further thought and research into this effect. However, the fact that these measurements are repeatable, that the heat rate is steady and uniform with time after about $100 - 150 \mu\text{sec}$., and that the heat rate at the base center varies like $p_1^{0.8}$, may be used as arguments to indicate that these effects are real.

The unexpected distribution of heat transfer rate over the base, particularly the high value at the center of the base, shows the necessity for careful study of wake phenomena. One way to attack this problem is to perform similar experiments in the shock tube with a wide range of model sizes. If the observed effect is caused by the unsteady flow conditions, the size of the model should affect the results. Because of the limitation in size of the present shock tube such a program could not be undertaken at this time. In order to shed more light on this problem, an investigation of base and wake flows is now underway in the GALCIT 5" continuous-flow hypersonic tunnel.

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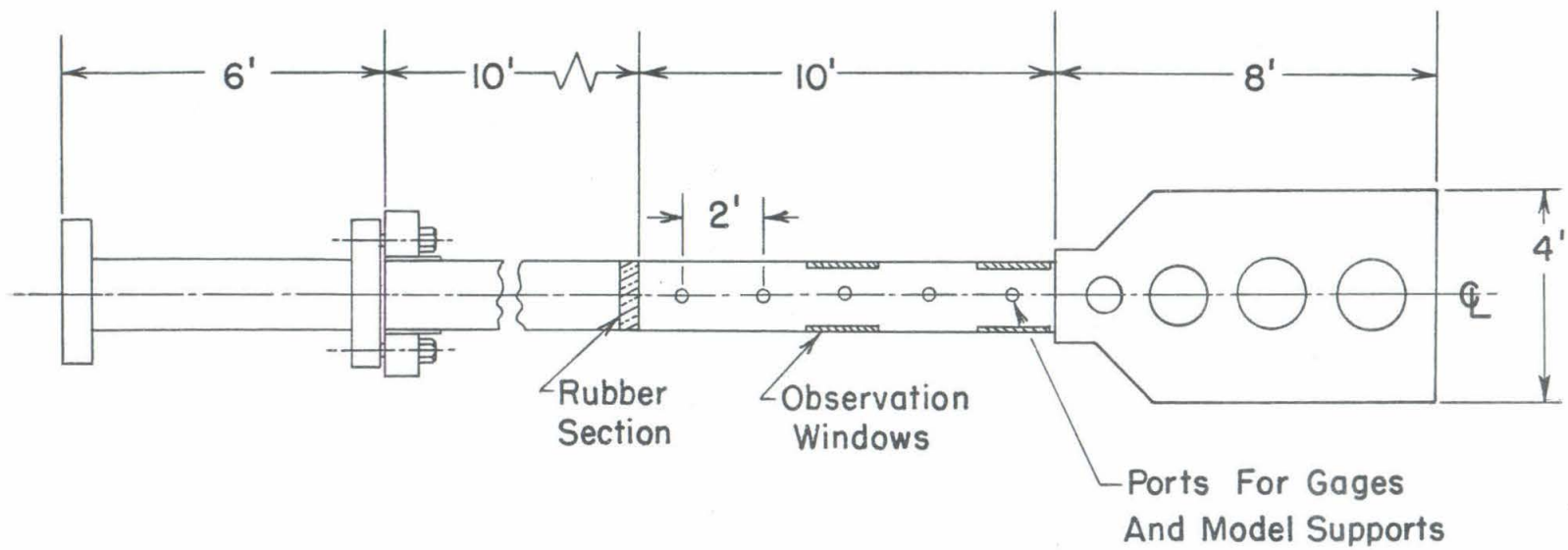


FIG. 1 - $2\frac{7}{8} \times 2\frac{7}{8}$ GALCIT SHOCK TUBE

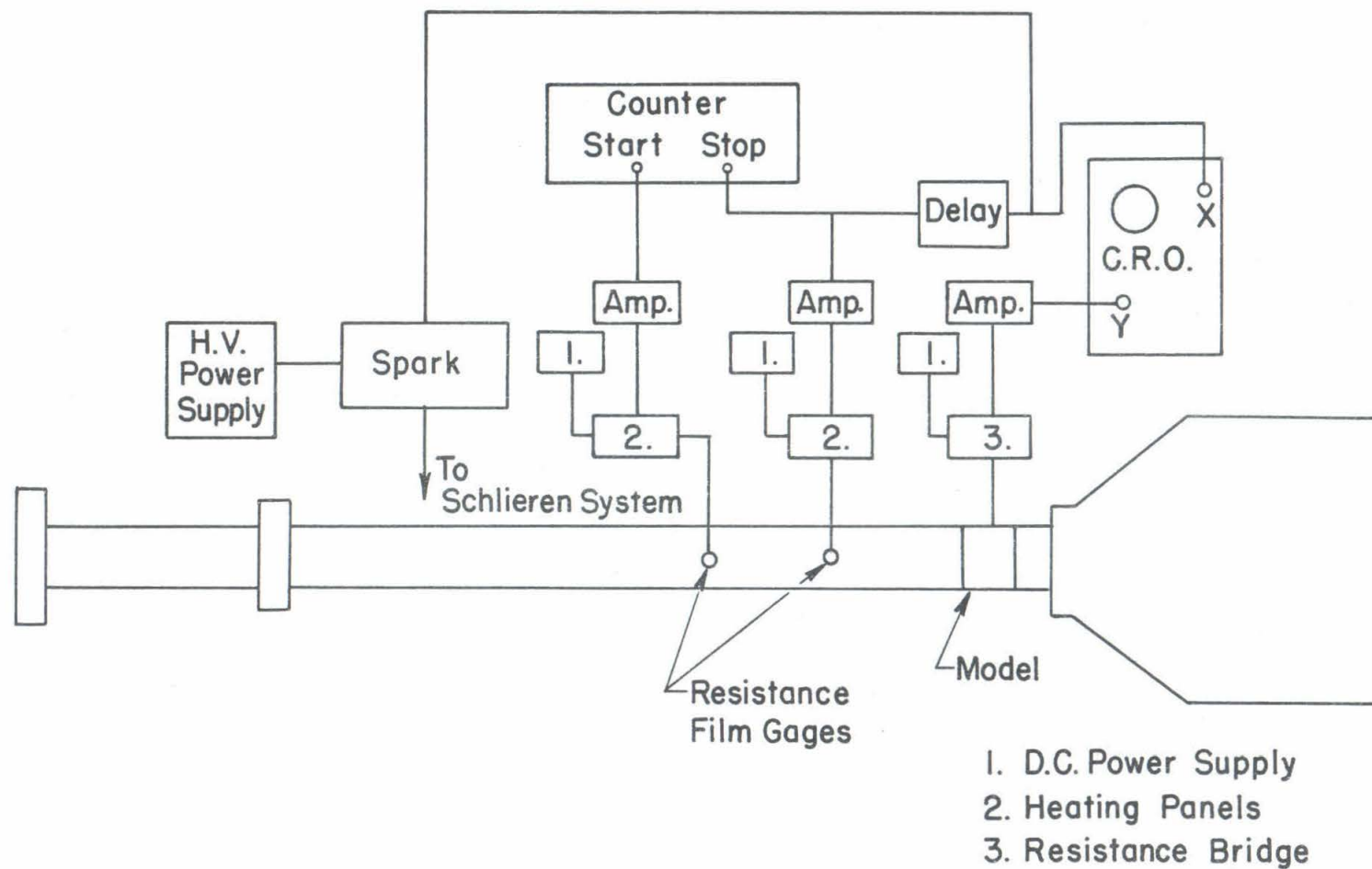


FIG.2 - INSTRUMENTATION BLOCK DIAGRAM



(a)

Side View



(b)

Back View

FIG. 3

HEMISPHERE CYLINDER MODEL

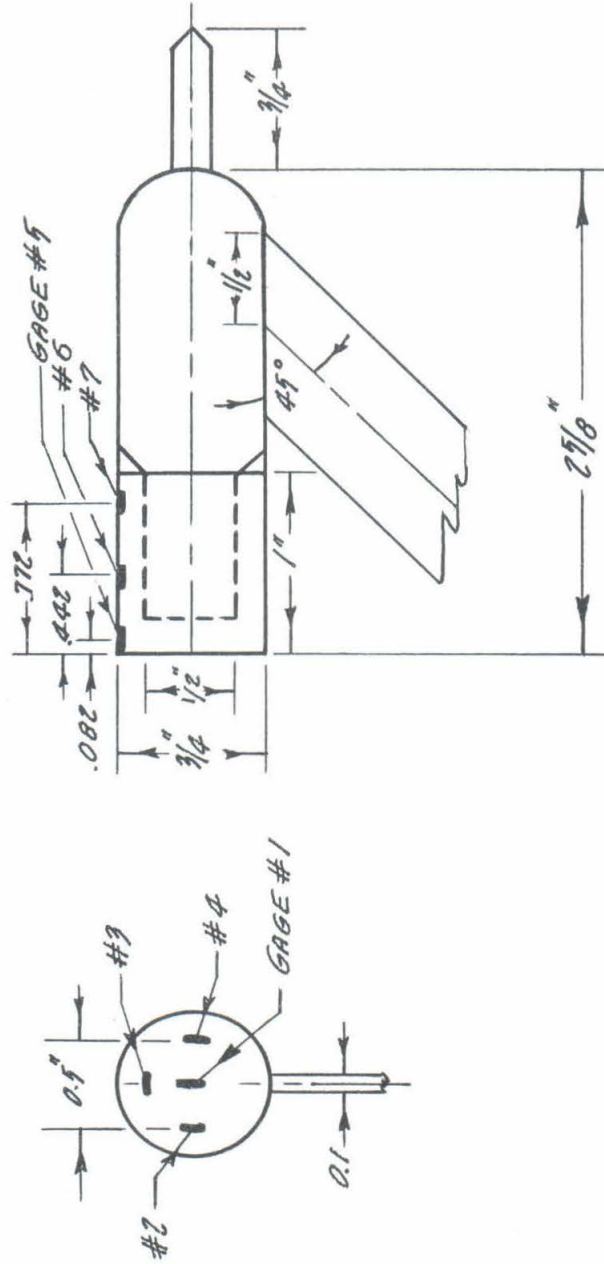
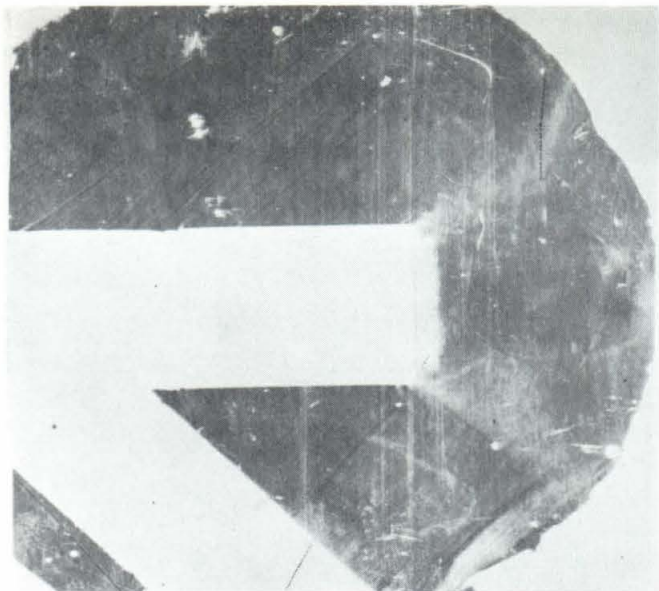
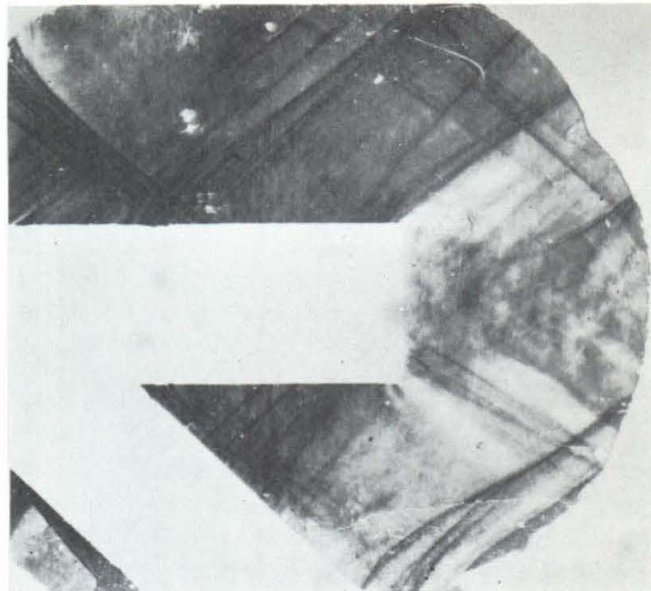


FIG. 4 DIAGRAM OF GAGE POSITIONS ON THE MODEL



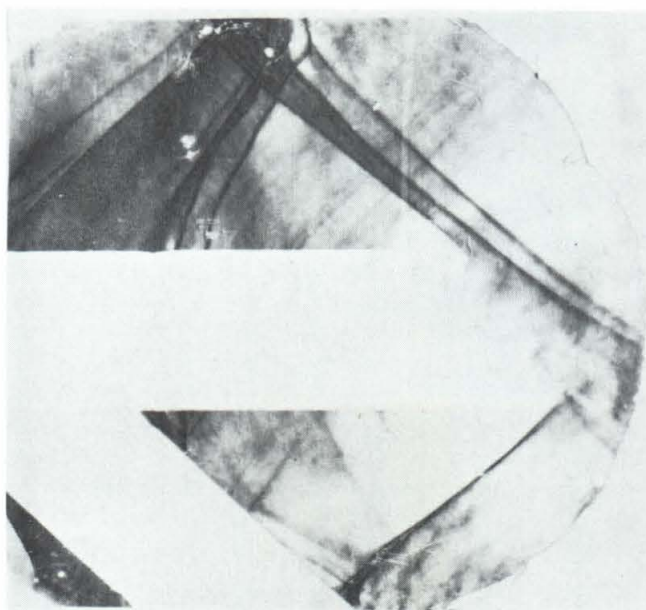
(a)

$M_s = 3.8$; $p_1 = 11.7$ cm. Hg.;
 $\Delta t = 40 \mu\text{sec.}$



(b)

$M_s = 3.8$; $p_1 = 11.3$ cm. Hg.;
 $\Delta t = 140 \mu\text{sec.}$

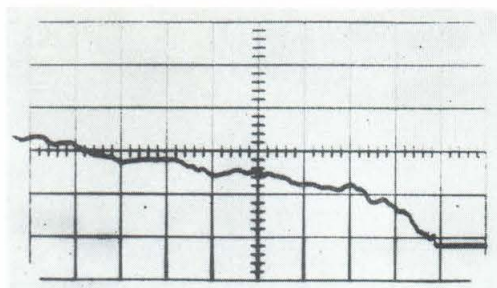


(c)

$M_s = 3.0$; $p_1 = 12.5$ cm. Hg.; $\Delta t = 360 \mu\text{sec.}$

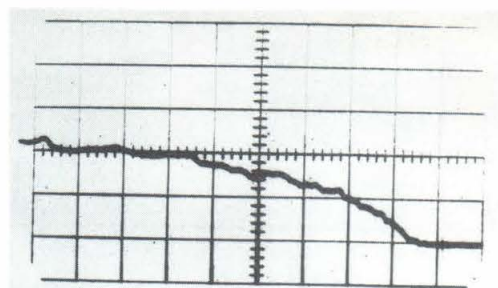
FIG. 5

SCHLIEREN STUDIES OF FLOW OVER THE AFT-PORTION OF THE MODEL



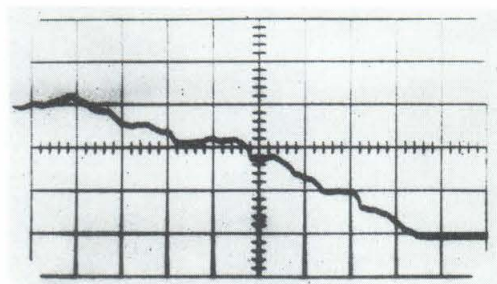
(a)

Gage No. 1

 $M_s = 3.5$; $p_1 = 8.0$ cm. Hg.Sweep = $100 \mu\text{sec./div.}$ $R_o = 24.9 \Omega$; $I_o = 20$ ma.Sensitivity = 10 mv./div.

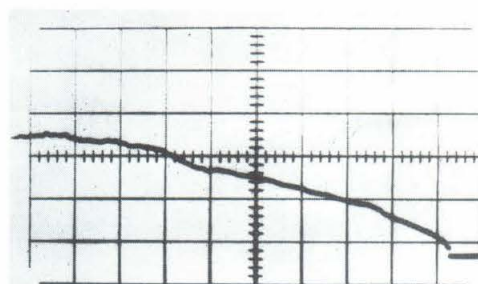
(b)

Gage No. 3

 $M_s = 3.6$; $p_1 = 6.6$ cm. Hg.Sweep = $100 \mu\text{sec./div.}$ $R_o = 19.0 \Omega$; $I_o = 20$ ma.Sensitivity = 5 mv./div.

(c)

Gage No. 4

 $M_s = 3.5$; $p_1 = 6.9$ cm. Hg.Sweep = $100 \mu\text{sec./div.}$ $R_o = 20.5 \Omega$; $I_o = 20$ ma.Sensitivity = 5 mv./div.

(d)

Gage No. 7

 $M_s = 3.81$; $p_1 = 6.86$ cm. Hg.Sweep = $100 \mu\text{sec./div.}$ $R_o = 19.3 \Omega$; $I_o = 20$ ma.Sensitivity = 10 mv./div.

FIG. 6

REPRESENTATIVE GAGE OUTPUT AT VARIOUS POSITIONS ON THE MODEL

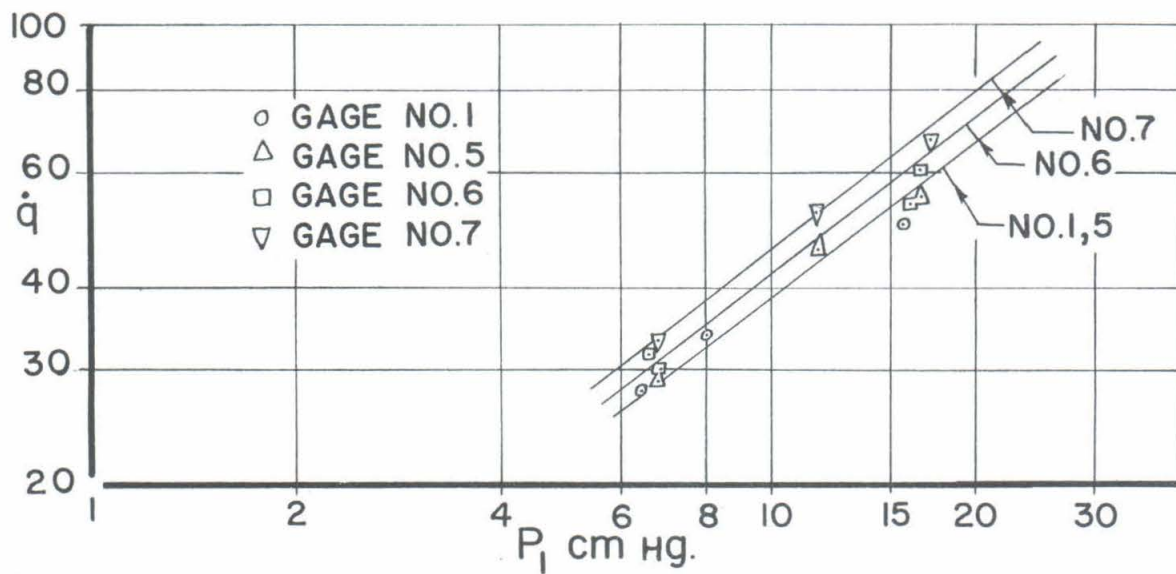


FIG.7 TURBULENT HEAT TRANSFER RATE VARIATION
WITH PRESSURE AT $(\bar{M}_s)=3.8$

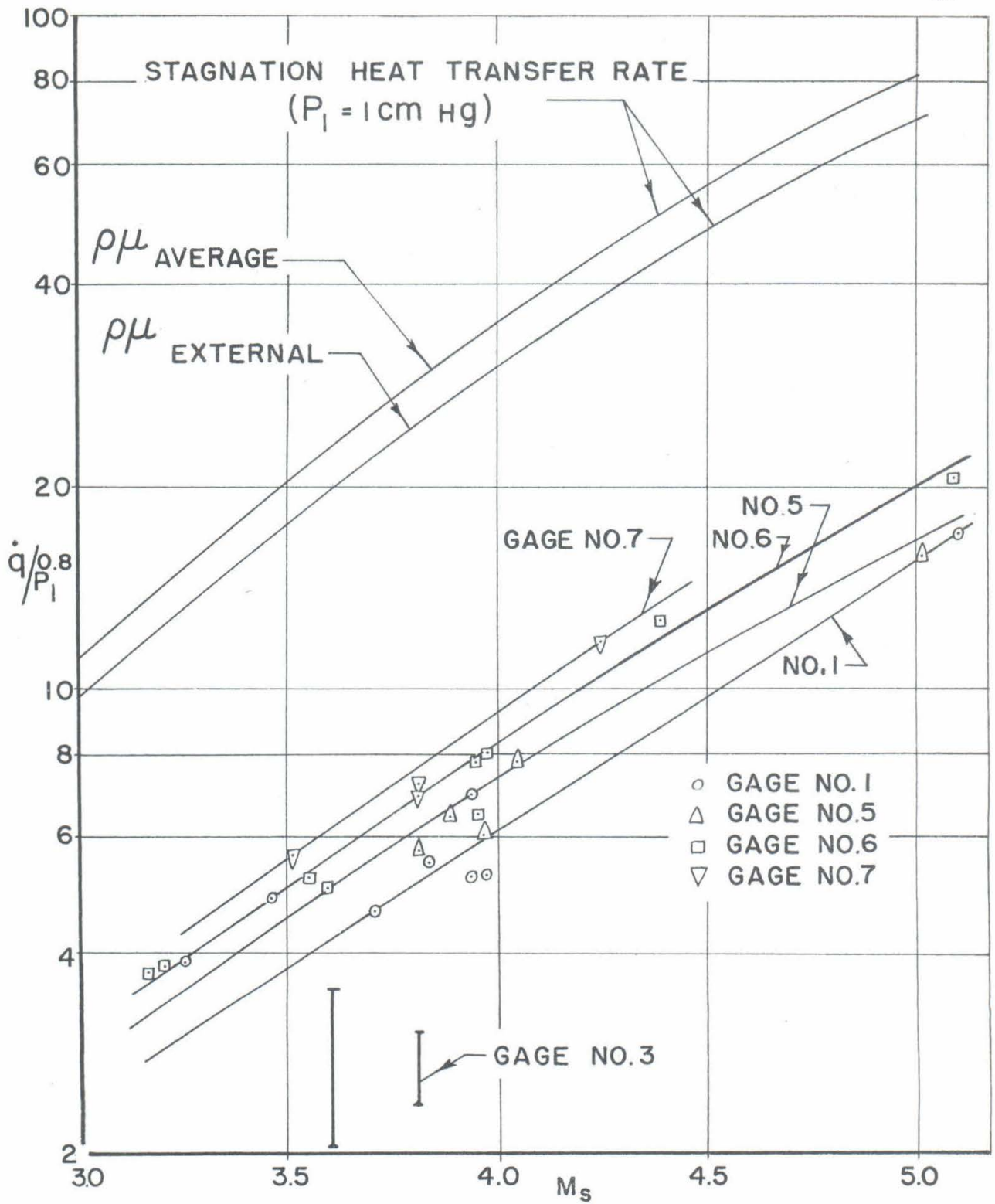


FIG. 8 TURBULENT HEAT TRANSFER RATE VARIATION
WITH SHOCK MACH NUMBER

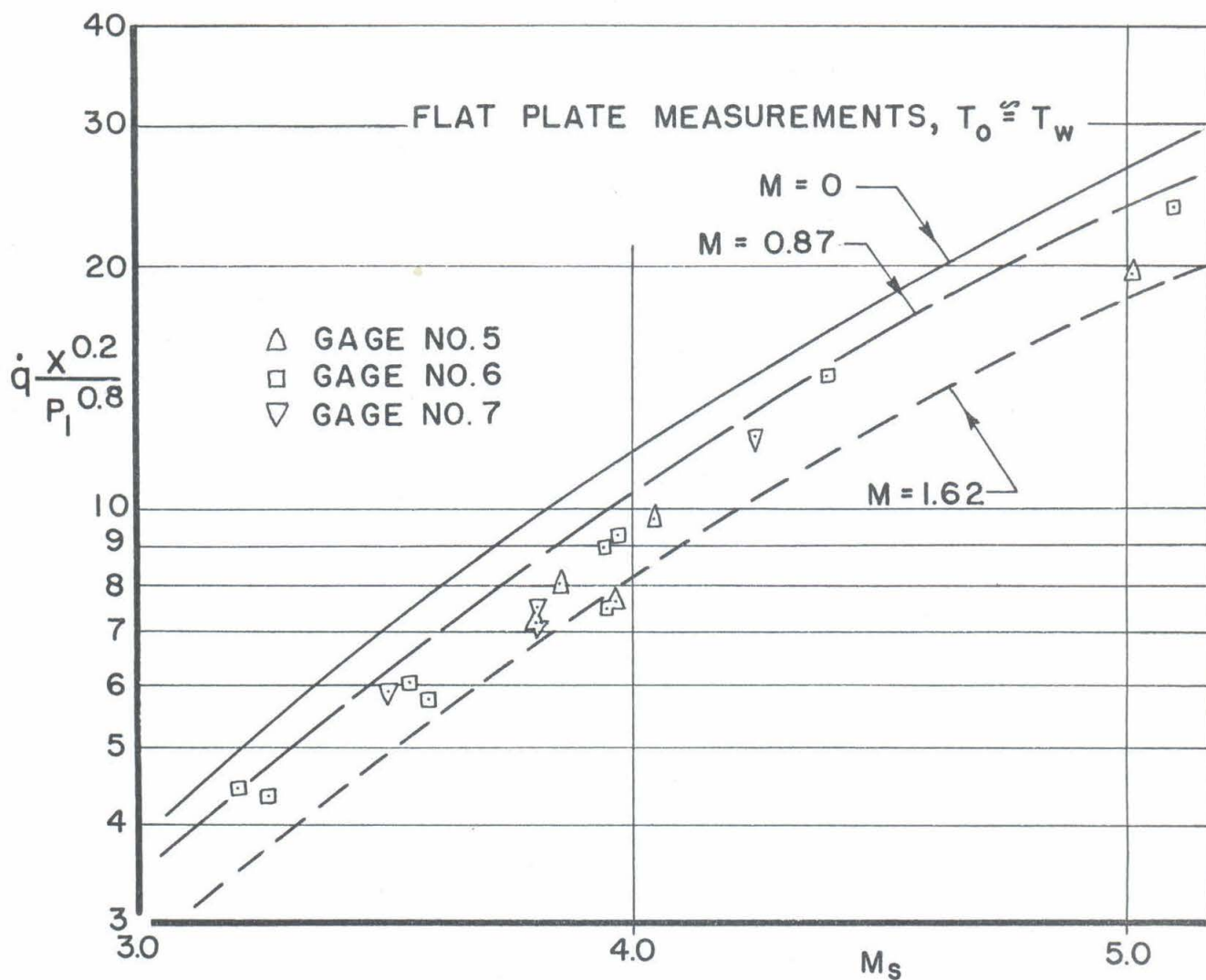


FIG.9 COMPARISON OF TURBULENT HEAT TRANSFER RATE WITH FLAT PLATE MEASUREMENTS

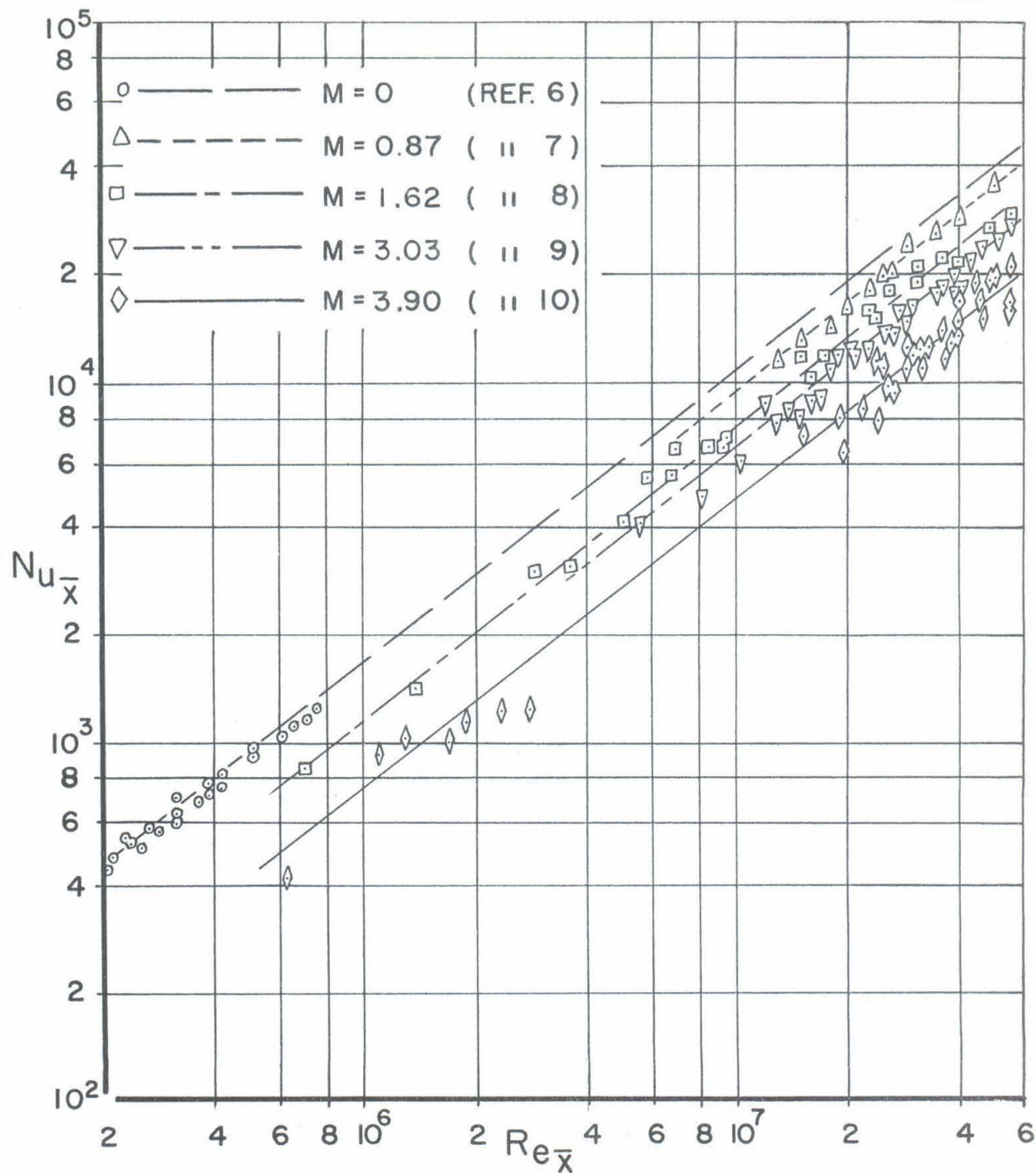


FIG.10 MEASUREMENTS OF TURBULENT HEAT
TRANSFER RATES ON A FLAT PLATE,
 $T_0 \approx T_W$ (REF. 6,7,8,9 AND 10)

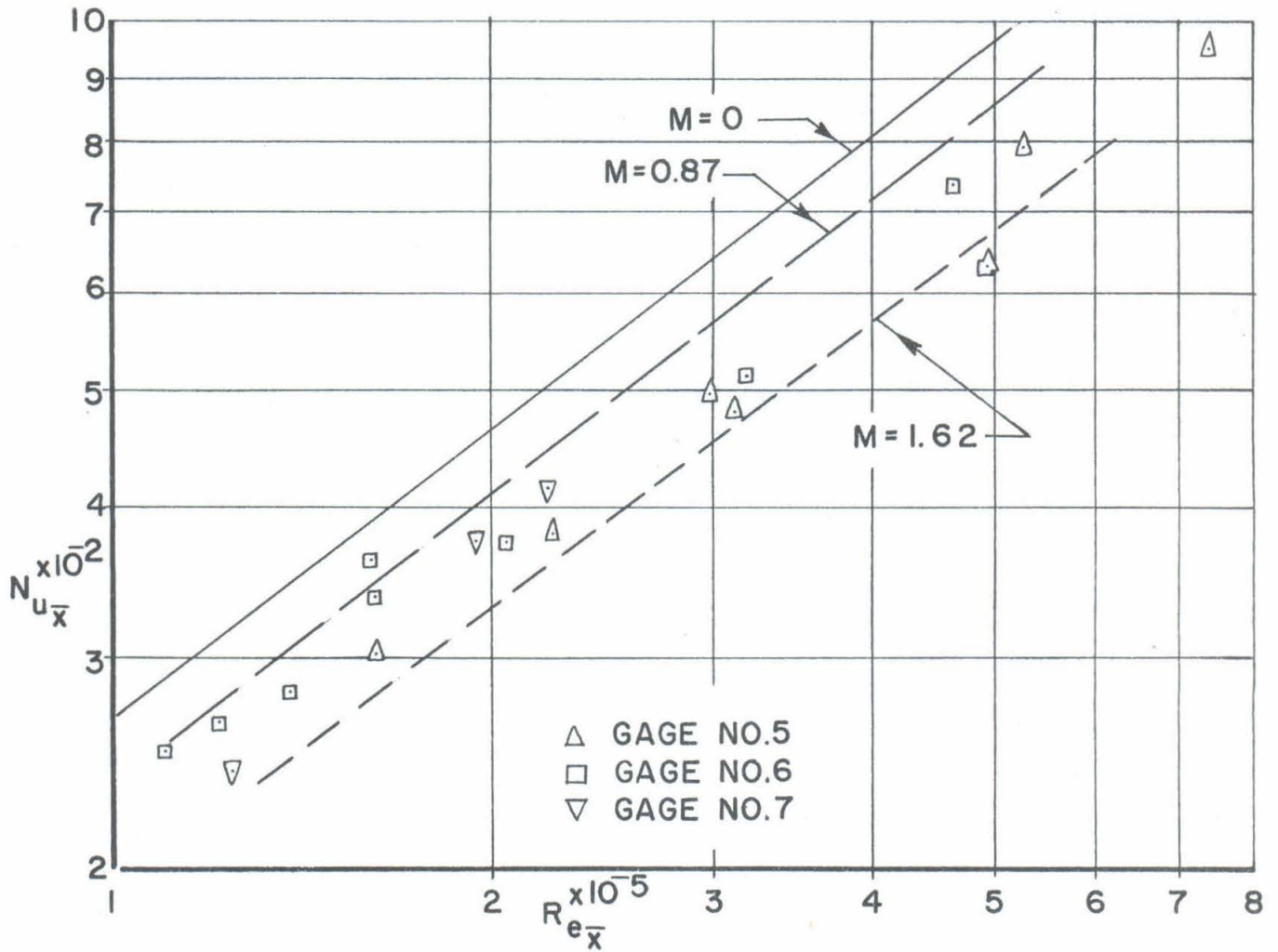


FIG.II COMPARISON OF THE SHOCK TUBE TURBULENT HEAT TRANSFER RATES WITH THE FLAT PLATE RESULTS

December 15, 1957

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